Exhibit I, 04/16/01
Contract No. xxxxxx

Mars Reconnaissance Orbiter

Orbiter Requirements

JPL D-20381

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1. Introduction

1.1. Purpose

The purpose of this document is to define the requirements on the MRO orbiter.

1.2. Document Structure

Section 1 provides an introduction to this document, definitions and applicable documents. Section 2 provides an overview of the mission. Section 3, 4, 5 and 6 contain requirements on the orbiter design from the following sources, project level, mission design, mission operations and payload accommodation.

1.3. Orbiter Implementation Overview

Science instrument requirements provided in this document are based upon a reference science payload. The final science payload will be selected via the NASA HQ Announcement of Opportunity process, which is not expected to be complete until late summer 2001.

In addition to the science instruments, the MRO payload includes 2 engineering elements; an UHF communications and navigation package called Electra and an optical navigation camera experiment.

The MRO launch vehicle will be in the intermediate class e.g. Delta III/IV, Atlas III/V and will be selected late in 2001 via the National Launch Service contract being managed by NASA KSC. MRO will be launched from Cape Canaveral Air Force Station.

In addition to the required payload elements; the Small Deep Space Transponder (SDST), a traveling wave tube amplifier (TWTA), and a Command Decoding ASIC will be available as GFP.

1.4. Definitions

1. Payload: The payload consists of the complement of the science instruments and engineering payloads that are provided as GFP.

2. Payload Mass: The payload mass is defined to include the GFP science and engineering payload [sensors, electronics and antennas] as well as the following items; booms, radiation shields, intra-instrument cabling and instrument multi layer insulation. Mounting hardware/brackets are excluded from the payload mass.

3. Orbiter: The orbiter is the composite of the engineering subsystems (commonly referred to as a “bus”) and includes the payload after integration. Included as a part of the orbiter is an adapter or other means, which provides the orbiter side of the interface with the Launch Vehicle.

4. Ground Support Equipment: The ground support equipment (GSE) comprises the ground hardware and software required to handle, verify and check out the elements of the orbiter during fabrication, assembly, testing, prelaunch, launch, and post-launch operations.

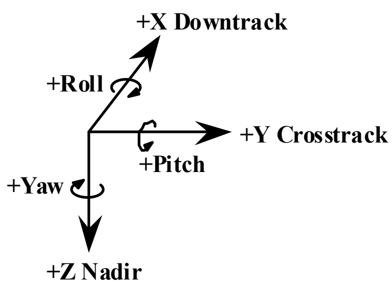
5. Mission Operations System/Ground Data System: The MOS/GDS consists of the complement of ground hardware, software, facilities, procedures, and personnel used to conduct flight operations.

6. DSN Pass: A DSN pass is defined as a single continuous 8-hour period during which a Deep Space Network (DSN) station is available for orbiter tracking, commanding, and telemetry return.

7. Solar Conjunction: The time period when the Sun-Earth-Mars angle is less than or equal to 5.0 degrees.

8. Quarantine Orbit: Near circular, 430-km altitude orbit.

9. Mars orbit reference coordinate system



Orbital Reference Coordinate System

The fundamental planet reference direction is the nadir direction. The +Z axis (nadir direction) is defined by a line passing through the orbiter perpendicular to the surface of the Mars reference spheroid (polar radius = 3375.7 km, equatorial radius = 3393.4 km). The +X axis is defined to lie in a plane perpendicular to the nadir direction and along the projection of the velocity vector on this plane. The +Y axis also lies in this plane and is orthogonal to both the +Z and +X axes, forming a right-handed coordinate system. As defined, the +X axis will be close to, but not always coincident with, the direction of the orbiter velocity vector; and the +Y axis will be close to, but not always coincident with, the orbit normal. "Yaw" shall be taken to refer to a rotation about the Z-axis, "roll" to a rotation about the X-axis, and "pitch" to a rotation about the Y-axis. Pointing control (in terms of roll, pitch, and yaw) shall refer to the +X, +Y, and +Z axes as defined above.

1.5. Applicable Documents

The following documents apply to the extent specified in this document, with revision number/date of issue as specified in Exhibit II, Applicable Documents.

Document No.	Title
EWR 127-1	Eastern and Western Range Safety Requirements
KHB 1710.2	Kennedy Space Center Safety Practices Handbook
JPL D-20327	MRO Mission Assurance Plan
JPL D-20241	MRO Preliminary Environmental Requirements and Estimates
JPL 810-005	Deep Space Network/Flight Projects Interface Design Handbook
JPL D-20393	Mission and Trajectory Description Document
JPL ES-518193	Small Deep Space Transponder Specification
JPL D-17868	JPL Design, Verification/Validation and Operations Principles for Flight Systems
NPG 8020.12B.	Planetary Protection Provisions for Robotic Extraterrestrial Missions
JPL D-20454	MRO Project Policies Document

2. Reference Mission Description

The Mars Reconnaissance Orbiter 2005 mission will deliver a single orbiter to Mars which will conduct remote sensing science observations, conduct site characterization for future potential landers, and provide a telecom/navigation relay capability for follow-on missions. The mission will be designed to provide global access from a low altitude orbit that is consistent with recovery of the MCO science objectives by the previously selected MCO investigations, and with additional high-priority science objectives and payloads defined by the MRO AO process.

The MRO orbiter will be launched and injected onto an interplanetary trajectory by an intermediate-class expendable launch vehicle (ELV) [for example, a Delta III/IV or an Atlas III/V]. The launch and injection will occur during the Mars opportunity of August 2005. The transit to Mars will have a duration of approximately seven months. During the course of the mission, the orbiter will communicate with the Earth via direct X band using NASA's Deep Space Network (DSN).

After arriving at Mars in March 2006, the MRO orbiter will be propulsively inserted into an initial, highly elliptical capture orbit with a period of 35 hours. The orbiter will then use aerobraking techniques to supplement its onboard propulsive capability and reduce its orbit period to that necessary for the primary science orbit (PSO). Aerobraking will consist of 3 distinct phases: a walk-in phase, a main phase, and a walkout phase. During the walk-in phase, the orbiter establishes initial contact with the atmosphere as the periapsis altitude of the orbit is slowly lowered. This phase continues until the dynamic pressures and heating rate values required for main phase, or steady state, aerobraking are established. During main phase, large-scale orbit period reduction occurs as the orbiter is guided to dynamic pressure limits. Main phase continues until the orbit lifetime of the orbiter reaches 2 days. (Orbit lifetime is defined as the time it takes the apoapsis altitude of the orbit to decay to an altitude of 300 km.) When the orbit lifetime of the orbiter reaches 2 days, the aerobraking walkout phase will begin. During the walkout phase, the periapsis altitude of the orbit will be slowly increased as the 2-day orbit lifetime of the orbiter is maintained. Once the orbit reaches an apoapsis altitude of 450 km, the orbiter will terminate aerobraking by propulsively raising the periapsis out of the atmosphere. Because the PSO has nodal orientation requirements, the aerobraking phase of the MRO mission must proceed in a timely manner and be completed near the time the desired nodal geometry is achieved.

Following the completion of aerobraking, the orbiter will perform a series of propulsive maneuvers to establish the PSO. The PSO for the mission will have a minimum periapsis altitude near 200 km and an apoapsis altitude near 400-km. The PSO will be near polar, Sun-synchronous with an ascending node orientation of 3:00 PM local mean solar time (LMST). Repetitive and targeted observations of the planet's surface and atmosphere will be conducted over a time span of one complete Martian year (687 Earth days), from November 2006 through November 2008. Throughout this time period, the orbiter will have its scientific instruments nadir pointed for surface observations. To enhance surface observation capability, the imaging instruments will acquire data from a cross-track orientation up to 30 deg from nadir. During the PSO, science data acquisition will be planned such that data can be downlinked to the DSN during two 8-hour 34m tracking passes every day.

Near the end of the primary science phase of the mission, MRO will support the Mars exploration program by providing approach navigation and relay communications support to various Mars landers and orbiters through its telecommunications/navigation subsystem. Additionally MRO will continue to serve as an asset for the Mars Program by continuing to use its science instruments to evaluate landing sites for future missions and extend global reconnaissance.

Mission Phase	**Duration**	**Description**
Launch	Several Hours	Extends from the start of the launch countdown to the initial acquisition, by the DSN, of the orbiter in a safe and stable configuration.
Cruise	About Five Months	Extends from orbiter separation to two months prior to the Mars Orbit Insertion (MOI) maneuver. It includes initial checkout of the orbiter and its payloads, any calibration or validation activities needed to ensure expected performance in subsequent mission phases, and required trajectory correction maneuvers (TCMs).
Approach and Orbit Insertion	About Three Months	Extends from two months prior to MOI, through MOI, and until the orbiter is checked out and ready to begin aerobraking. The orbiter is inserted into a nearly polar orbit with a period of 35 hours.
Aerobraking	four to Six Months	Begins after Mars orbit insertion and on-orbit checkout and ends prior to the start of primary science. During this phase the orbiter's orbit periapsis is lowered to the upper reaches of Mars' atmosphere. The apoapsis of its initial capture orbit is aerodynamically reduced to give an orbit period of about 2 hours.

Primary Science	Two Years	Begins after achieving the primary science orbit and continues for one Mars year (687 Earth days) after science instrument turn-on. The orbiter will provide relay support to Mars missions launched in 2007.
Relay	About Two Years	Begins with the arrival of the first spacecraft in the international Mars exploration program beyond MRO and continues for the remainder of the orbiter's on-orbit life. During this phase, priority will be given to the communications relay/navigation support functions for other spacecraft in the international Mars exploration program.

3. Project Requirements

3.1. Lifetime

The orbiter shall be designed to operate within specification for 5.4 years after launch.

The orbiter propellant loading shall enable 10 years of mission life.

3.2. Planetary Protection

The orbiter shall comply with the Planetary Protection requirements for a class III, "C" mission as defined in Planetary Protection Provisions for Robotic Extraterrestrial Missions, NPG 8020.12B.

3.3. Single Point Failures

The orbiter shall not contain single point failures except for those allowed by exemption in the MRO Project Policies document.

The interface between Electra and the orbiter shall not contain single point failures.

3.4. Use of Radioisotope Materials

No radioisotope materials shall be used; this includes, but is not limited to, Radioisotope Thermal-electric Generators (RTGs), and Radioisotope Heating Units (RHUs).

3.5. Design Principles

The orbiter shall be designed to be compliant with JPL D-17868 "Design, Verification/Validation and Operations Principles for Flight Systems". Exceptions to JPL D-17868 shall be justified with detailed rationale. MRO project allowed exceptions are documented in the MRO Project Policies document.

3.6. *Testability*

The orbiter shall be tested in accordance with the provisions of JPL D-20241, "MRO Preliminary Environmental Requirements and Estimates".

3.6.1. Orbiter Testability

Critical functions of the orbiter and interactions between the orbiter and the MOS, launch vehicle and DSN shall be testable at a level of assembly adequate to verify the function or interface to be tested. Normal and fault protection/correction hardware and software states and sequences shall be testable at the system level on the flight vehicle. The orbiter and test system shall provide for safe ground testing of all hazardous orbiter commands.

3.6.2. Government-Furnished Property Testability

The orbiter design shall provide for functional verification of the payload operation at the orbiter level. Hardware and software interactions among GFP elements and between GFP elements and the orbiter shall be testable at the system level. Access to and electrical monitoring of payload interfaces shall be possible at the system level. Access for external stimulation of payload sensor elements shall be possible at the system level. Payload interfaces shall be designed so as not to limit installation/removal cycles or test duration at the system level.

3.7. Orbiter Environments

The orbiter shall be designed to meet the requirements as specified in this document when operating in the expected mission environments described in JPL D-20241 “MRO Preliminary Environmental Requirements and Estimates” which will be superceded by the contractor created Environmental Requirements Document.

Environmental test and design margins are specified in Appendix A of, in JPL D-20241 “MRO Preliminary Environmental Requirements and Estimates”.

4. Mission Requirements

4.1. Launch Period

The orbiter shall be capable of launch on any day during the period Aug 8, 2005 - Aug 28, 2005.

The orbiter shall be capable of supporting a 21-day launch period.

4.2. Daily Launch Opportunities

The orbiter shall be capable of supporting a 2-hour daily launch window.

4.3. Launch Mass

The maximum launch mass of the orbiter shall be 1800 kg.

4.4. Launch Vehicle

The orbiter shall be compatible with a NASA provided intermediate class [e.g. Delta III/IV, Atlas III/V] launch vehicle.

4.5. Launch Vehicle Integration

The orbiter shall integrate with the launch vehicle at the Cape Canaveral Air Station per the approved Launch Site Support Plan.

4.6. Launch Site Safety

While at the launch site, the orbiter shall comply with the requirements of the Range Safety Document EWR-127 and KHB-1710.2.

4.7. Trajectory Correction Maneuvers

The orbiter shall be capable of performing the first trajectory correction maneuver (TCM) anytime after launch plus 8 days and shall be capable of performing at least 5 TCMs.

The last TCM is targeted not closer than 12 hours before Mars Orbit Insertion (MOI).

The orbiter shall be capable of executing a TCM in any inertial direction, subject to the accommodation of payload sun, time-of-travel requirements as negotiated in the payload ICDs.

4.8. Arrival Dates

The orbiter shall be capable of arriving on any day during the period Mar 3, 2006 to Mar 11, 2006.

4.9. MOI Direction

The orbiter shall be designed to support a Southern approach Trajectory to Mars for the MOI maneuver.

4.10. Aerobraking

The orbiter shall support aerobraking for lowering the capture orbit apoapsis to an altitude of 450 km..

The orbiter shall be capable of supporting 180 days of sustained aerobraking operations.

During aerobraking, the orbiter shall support, on any orbit, daily maneuvers designed to adjust the periapsis altitude.

The orbiter shall be designed to support a 2-day orbit lifetime requirement while aerobraking. [Note: The orbit lifetime is defined as the time necessary for apoapsis altitude to decay to an altitude of 300 km using a nominal atmospheric model.]

4.11. Aerobraking Contingency Pop-up Maneuvers

The orbiter shall be capable of performing at least one contingency pop-up maneuver to raise periapsis out of the atmosphere.

4.12. Aerobraking Drag Pass Detection

The orbiter shall be capable of measuring and recording the integrated (accumulated) deceleration in 1 second intervals (counts) with a maximum deceleration of 100 mm/sec/count, a quantization of 0.01 mm/sec/count and an accuracy of 0.05 mm/sec/count (3 sigma, including errors due to bias and drift).

The orbiter shall be capable of measuring and recording the attitude and attitude rates in 1 second intervals.

The orbiter shall be capable of returning the deceleration data to the ground after each aerobraking pass.

4.13. Primary Science Orbit

The orbiter shall have the capability to propulsively establish the primary science orbit after aerobraking to an apoapsis alititude of 450 km.

The orbiter shall achieve the primary science orbit at least 14 days prior to the start of solar conjunction.

The orbiter shall be capable of operating within specification in the primary science orbit described by the following parameters.

a) nodal orientation at 3:00 PM [+/- 15 minutes] local mean solar time (LMST)
b) a nodal orientation that is Sun-synchronous
c) a minimum periapsis radius of 3575 km
d) a maximum periapsis radius of 3745 km
e) an average* apoapsis radius of 3794 km or less
(*due to gravity field perturbations over 60 days)

The reference primary science orbit is specified in the following table.

Parameter	Value	Required Range/Characteristic
Apoapsis Radius, Ra , km	3790.8	Average Ra <= 3794 km
Periapsis Radius, Rp ,km	3625.5	Minimum Rp = 3575 km, Maximum Rp = 3745 km
Inclination, i, deg	92.815	Consistent with: Sun-synchronous f(Ra, Rp, i)
Ascending Node, Ω , deg	-23.975	Consistent with: 3:00 PM +/- 15 minutes LMST
Argument of Periapsis, ω , deg	-2.286	Variable: 0 to 360 deg
Mean Anomaly of Epoch, deg	0.000	Variable: 0 to 360 deg
Orbit Epoch, t_0	6/21/2006 01:48:48.817 Ephemeris Time	
Coordinate Reference	Mars Mean Equator and IAU-Vector of Epoch	

4.14. Non-Gravitational Accelerations

4.14.1. Solar Pressure

During all mission phases, the orbiter solar pressure estimate uncertainty due to orbiter design and construction shall be less than 10% (3-sigma).

4.14.2. Orbiter –Generated Accelerations

The orbiter shall be designed such that the thrust uncertainty due to momentum management shall be less than 0.3 mm/s (3-sigma) per axis per event. This requirement applies to predicted values up to 10 days in advance.

The orbiter shall be designed such that the momentum management events shall occur no more frequently than once every 48 hours. Momentum management events shall be performed during telecommunication with the Earth.

The orbiter shall be designed such that the self-generated translational accelerations imparted by outgassing and propellant leaks shall be less than $3x10^{-13}$ km/s^2 (3-sigma)

It is not required to meet the orbiter-generated acceleration requirements during emergency momentum events, aerobraking, planned or unplanned thruster slew events, or while the orbiter is in safe hold.

4.15. Orbiter Ephemeris in the Primary Science Orbit

4.15.1. Ephemeris Prediction.

Orbiter functions dependent on orbit ephemeris predictions shall assume the performance given in Table 4.15-1.

Table 4.15-1 Primary Science Orbit Ephemeris Prediction Errors (3-sigma) (Position errors at 7 days past DSN tracking data OD cutoff)

	Radial (km)	Downtrack (km)	Crosstrack (km)
Position Error	0.04	1.5	0.05

Note: The ephemeris predication capability assumes that the orbiter design meets the requirements of Section 4.14.

4.15.2. Ephemeris Reconstruction.

Orbiter functions dependent on orbiter ephemeris reconstruction shall assume the performance given in Table 4.15-2.

Table 4.15-2. Primary Science Orbit Ephemeris Reconstruction Errors (3-sigma)

	Radial (km)	Downtrack (km)	Crosstrack (km)
Position Error	0.01	0.3	0.04

Note: The ephemeris predication capability assumes that the orbiter design meets the requirements of Section 4.14.

4.16. Orbit Maintenance

The orbiter shall be capable of performing orbit trim maneuvers (OTMs) as frequently as every 7 days. The orbiter shall be capable of executing an OTM velocity increment in any inertial direction.

4.17. Relay Phase

The orbiter shall be capable of performing relay phase maneuvers as frequently as every 7 days. The orbiter shall be capable of executing a relay phase velocity increment in any inertial direction.

4.18. Quarantine Orbit

At the end of the mission, the orbiter shall be capable of maneuvering to the quarantine orbit.

4.19. Total Required Translational Delta-V

The orbiter shall be capable of providing the required translational delta-Vs shown in the following table. The orbiter shall provide additional delta V to accommodate finite burn losses and orbiter attitude maneuvers for all mission phases.

Required Translational Delta-V Budget

Required Trajectory Correction	**Direct (m/s)**	**Comments**
Total Earth-Mars TCM delta-V	50	
Mars Orbit Insertion	1035	35 hr orbit, 3700 km periapsis radius
Capture Orbit Trim	10	
Aerobraking	53	Multiple burns none greater than 20 m/s nor less than 0.05 m/s
Aerobraking termination	46	
Orbit Adjust 1	41	Go from 450X200 to 400X200km plus inclination trim
Orbit Adjust 2	58	Go from 400X200 to 400X350km plus inclination change

Orbit maintenance	40	
Orbit Adjustment for Relay Ops	20	
Planetary Protection Quarantine Burn	30	To raise orbit from 350x400 to 430 circular
Total translational delta-V	**1383**	

4.20. Maneuver Execution Errors

The orbiter shall produce maneuver execution errors less than those specified the following table.

Manuever Execution Errors (3-sigma)

Fixed Magnitude Error (m/s)	Proportional Magnitude Error (%)	Fixed Pointing Error, per axis (m/s)	Proportional Pointing Error, Total(%)
0.02 m/s	± 2 %	0.02 m/s	± 2 %

5. Operability

5.1. Orientation During Primary Science and Relay Phase

During the primary science and relay phases, the orbiter shall be capable of continuously and autonomously maintaining the payload orientations in section 6.4 to the pointing specifications in section 6.15 except during maneuvers .

5.2. Telecommunications and Navigation Tracking Schedule

The orbiter shall be capable of supporting the telecommunications and navigational tracking schedule shown in the following table while simultaneously meeting the orientation requirements in 5.1.

Mission Phase	Key Events	Begin	End	Tracking Requirements
Launch		L + 000 days	L + 030 days	Continuous 34m coverage; 1 & 2 way coherent doppler; ranging; commanding and telemetry
	TCM 1	L + 015 days		Continuous 34m coverage; 1 & 2 way coherent doppler; ranging; commanding and telemetry
Cruise		L + 030 days	M - 060 days	One 8 hr pass per day 34m; 1 & 2 way coherent doppler; ranging; commanding and telemetry
	TCM 2	L + 090 days		+/- 3 days continuous 34m; 1 & 2 way coherent doppler; ranging; commanding and telemetry
	TCM 3	L + 120 days		+/- 3 days continuous 34m; 1 & 2 way coherent doppler; ranging; commanding and telemetry

Mars Approach and Orbit Insertion		M - 060 days	M + 030 days	Continuous 34m coverage; 1 & 2 way coherent doppler; ranging; commanding and telemetry
	Delta-DOR	M – 059 days		2 antenna at overlap, additional to continuous coverage; once per week until MOI-40 days; alternate East/West and North/South overlaps
	TCM 4	M – 020 days		Continuous 34m coverage; 1 & 2 way coherent doppler; ranging; commanding and telemetry
	TCM 5	M – 004 days		Continuous 34m coverage; 1 & 2 way coherent doppler; ranging; commanding and telemetry
	MOI	M - 000 days		70m coverage; one pass on either side of MOI
Aerobraking		M + 030 days	M + 270 days	
	Aerobraking	M + 030 days	M + 210 days	Continuous 34m coverage; 1 & 2 way coherent doppler; ranging; commanding and telemetry
	Transition to Mapping	M + 210 days	M + 270 days	Two 8 hr passes per day 34m; 1 & 2 way coherent doppler; ranging; commanding and telemetry
	Solar Conjunction	M + 218 days	M + 250 days	One 8 hr pass per day 34m; 1 & 2 way coherent doppler; ranging; commanding and telemetry
High Resolution Mapping		M + 270 days	M + 1000 days	Two 8 hr passes per day 34m; 1 & 2 way coherent doppler; ranging; commanding and telemetry
Relay		M + 1000 days	M + 1765 days	One 8 hr pass per day 34m; 1 & 2 way coherent doppler; ranging; commanding and telemetry

5.3. Science Data Characterization

The orbiter shall provide engineering data for incorporation in the telemetry stream that enables correlation of the orbiter's state, performance, attitude and environment with the science data.

5.4. Engineering Telemetry

The orbiter shall provide sufficient time tagged engineering telemetry to define an unambiguous orbiter state during nominal and emergency conditions, and after executing a real-time or stored sequence command.

The uplink data rate capable of being received by the orbiter shall be part of the engineering telemetry.

5.5. Autonomous Operations

The orbiter shall provide the capability to autonomously manage/perform the following functions.

1) Ephemeris relative commanding
2) Orbiter momentum
3) Data storage , retrieval and deletion
4) Accommodate sun time-of-travel requirements from the payload during attitude maneuvers
5) Launch
6) Mars Orbit Insertion
7) Unattended operations for 28 days during normal operations

5.6. Operation During Solar Conjunction

The orbiter shall be able to function without ground commands and maintain the minimum functions required for safe system operation during solar conjunction.

5.7. Command and Telemetry Accessibility

Once separated from the launch vehicle, the orbiter shall be capable of continuously receiving commands and transmitting real time and stored telemetry**.**

5.8. Clock Correlation

The orbiter shall maintain synchronicity with the ground clock i.e. UTC to within 0.1 seconds and shall require no more than 1 update per week.

The correlation between the orbiter clock and the ground clock shall be capable of being reconstructed to an uncertainty of less than 15 msec.

5.9. Communications

5.9.1. DSN Compatibility

The orbiter shall be compatible with the Deep Space Network (DSN) at X band as described in the DSN Interface Document JPL 810-005 and the tracking schedule as described in section 5.2.

The orbiter uplink and downlink shall be compatible with the 34m BWG and HEF antennas for nominal operations.

The orbiter uplink and downlink shall be compatible with the 70m antennas for emergency operations and enhanced science.

The orbiter uplink and downlink shall be compatible with the acquisition aid antennas for initial acquisition.

5.9.2. Transponder

The orbiter shall accommodate the GFP Small Deep Space Transponder [JPL ES-518193].

The orbiter shall support the DOR tone capability in the SDST.

Note: The SDST provides the capability to select either convolutional coding (K=7, Rate =1/2) or no coding.

5.9.3. Uplink

The orbiter shall have the capability to receive an X-band carrier, modulated with command and ranging subcarriers.

The orbiter shall be capable of providing a primary uplink path sufficient for simultaneous commanding and two-way coherent Doppler and range tracking at maximum Earth range.

5.9.4. X Band Channel Allocation

The orbiter shall use DSN Channel TBD for X-band uplink (TBD MHz) and downlink (TBD MHz).

5.9.5. Uplink Data Rate

The orbiter shall be capable of receiving commands via X band at 7.8125 bps (emergency), 2000bps (operational) and multiple intermediate rates.

5.9.6. Downlink

The orbiter shall have the capability to transmit an X band carrier modulated with engineering, navigation and science data.

5.9.7. Downlink Data Rate

The orbiter shall be capable of a minimum downlink data rate of 40 bps to a 34m BWG DSN antenna during normal cruise.

The orbiter shall be capable of an emergency downlink rate of 40bps at maximum Earth range to a 70m DSN antenna.

The orbiter shall be capable of a minimum downlink rate of 280 kb/s to a 34m BWG DSN antenna at the maximum Earth range of 2.67 AU during nominal operations.

The orbiter shall be capable of a maximum downlink rate of 4.4 Mbps to a 70m DSN antenna with no encoding, 4.4 MSps with 7 1/2 convolutional encoding, and 4.4 MSps with turbo coding (rate 1/3 and 1/6) both to a 34m BWG DSN antennas.

The orbiter shall be capable of multiple intermediate downlink rates between the minimum and maximum rates.

5.9.8. Bit Error Rate

The orbiter to DSN maximum Bit Error Rate (BER) shall be 1 x 10-5 for uplink and 1 x 10-6 for downlink.

5.9.9. External Oscillator

The orbiter shall be capable of using the USO frequency reference in the Electra package in both the command and telecommunications subsystems.

5.9.10. Orbiter Contribution to Doppler Measurement

The orbiter contribution to the X-band 1-way or 2-way doppler signal, exclusive of center of mass motion, shall not be greater than 0.1 mm/sec (1σ) over a sixty second integration time, except during momentum management periods.

5.10. *Command – Telemetry Design*

5.10.1. Data Format

The orbiter shall conform to the Consultative Committee for Space Data Systems (CCSDS) standards for telemetry and command data formats.

5.10.2. Command Accountability/Verification

The orbiter shall be capable of ensuring command integrity prior to execution and capable of reporting sufficient information to the ground operators to assess verification of command receipt and execution.

5.10.3. Command Processing Capability

The orbiter shall be capable of receiving and processing stored sequence command files and real time commands.

A stored sequence command file shall be associated with a time tag. The time tags shall be either an absolute execution time or a time relative to the execution of the previous command or command file. Time tags shall be sized to encompass the duration of the mission.

The orbiter shall be capable of running concurrent sequences of command files.

The orbiter shall be capable of terminating a command file if a fault is encountered or by ground command.

5.10.4. Telemetry Coding

The orbiter shall provide for either Reed-Solomon encoding, turbo (rate 1/3 and 1/6) encoding or no coding prior to sending telemetry to the SDST.

The orbiter shall transmit Reed-Solomon encoded telemetry with either convolutional coding or no coding.

The orbiter shall transmit turbo-coded telemetry with no additional coding.

5.10.5. Telemetry Prioritization

The orbiter shall support prioritization of telemetry packets. The prioritization shall be capable of being configured via ground command.

5.11. Fault Protection

5.11.1. Unattended Operation

In the event of a fault condition, the orbiter shall be capable of surviving autonomously for 14 days except during aerobraking.

5.11.2. Emergency Uplink

The orbiter shall be capable of accepting the emergency uplink signal following detection of a fault condition.

5.11.3. Emergency Downlink

The orbiter shall transmit the emergency downlink signal with Reed-Solomon and convolutional coding following detection of a fault condition.

The orbiter shall provide for autonomous initiation of emergency telemetry in the event that no commands are received within a ground-selectable time period

5.11.4. **Safe Mode**

During all mission phases following separation from the launch vehicle, including aerobraking, the orbiter shall enter a safe, and maintainable configuration and attitude following detection of a fault condition or by ground command.

The ability to inhibit safe mode entry shall be provided for critical events such as Launch, MOI, and OTM.

While in safe mode, the orbiter shall provide emergency uplink and downlink capability.

Orbiter telemetry prior to and after the safing event shall be stored for downlink upon ground contact.

The orbiter shall be capable of issuing a safing command to the payload and disable the payload prior to entering safe mode.

5.11.5. Safe Mode Exit

The orbiter shall exit safe mode by ground command only.

The orbiter shall return to normal operations within 6 hours after being commanded out of safe mode.

5.11.6. Critical Memory Protection

The orbiter shall provide protection from the adverse effects of single stuck bits or SEU faults for memory used to store flight code or other critical data.

6. Payload Requirements

6.1. Payload List

The orbiter shall accommodate the payload consisting of:

1) PMIRR MK II
2) MARCI+
3) High Resolution Imager
4) Visible-Near Infrared Imaging Spectrometer
5) Shallow subsurface sounding radar
6) Electra UHF/Navigation Package
7) Optical Navigation Camera Experiment

Details of the mechanical, thermal, electrical, and data interfaces will be contained in the orbiter to payload Interface Control Documents (ICDs).

The orbiter shall accommodate the RFI environment for Electra as negotiated in its ICD.

Note: Science instrument requirements provided in this document are based upon a reference science payload. The final science payload will be selected via the NASA HQ Announcement of Opportunity process, which is not expected to be complete until late summer 2001.

6.2. Volume and Mass

The orbiter shall accommodate a payload mass of 140 kg including all margins. The orbiter shall accommodate the following volumes for the science instruments. These volumes will be finalized in the instrument ICDs.

Note: The current best estimate (CBE) of the mass of Electra and the Optical Navigation Camera experiment is 15 and 3 kg, respectively.

Payload Element	Dimensions (cm)
PMIRR MK II	30 x 30 cm (diameter x height)
MARCI+	
Wide Angle Optics	6 x 7 (diameter x height)
Medium Angle Optics	12 x 30 (diameter x height)
Electronics	35 x 12 x 12 & 6 x 9 x 13
High Resolution Imager	
Optics	60 x 150 (diameter x height)
Electronics	24 x 24 x 6
Visible-Near Infrared Spectrometer	
Optics	70 x 30 x 30
Electronics	30 x 20 x 15
Scan Mirror Driver	13 x 13 x 3 cm
Shallow subsurface sounding radar	
Receiver	22 x 25 x 20

Transmitter Antenna box Deployed Dipole Antenna	70 x 25 x 5 45 x 25 x 10 700 tip to tip
Electra 2 Transceivers, each 2 USOs, each 2 UHF Antennas, each 1 X band LGA	 22 x 17 x 14 12 x 10 x 6 25 x 25 x 1 TBD
Op Nav Camera	40 x 21 x 22

6.3. Power

After achieving the primary science orbit, the orbiter shall be capable of providing an orbital average payload power of 200W including all margins for the payload elements.

While in cruise to Mars, the orbiter shall be capable of providing 80W including all margins for the payload elements.

The orbiter shall provide unregulated 28Vdc (+8Vdc, -6Vdc) power at the payload connectors. Each payload element shall be allocated one single fault tolerant power switch to turn power on and off. The Ultra Stable Oscillators (USOs) shall be separately switched as well as the Electra receivers. A separate switch shall be provided for the replacement heater power circuits in each payload element.

The orbiter shall provide the capability to turn on payload elements on the launch pad after integration to the launch vehicle.

6.4. Field of View and Orientation

The orbiter shall accommodate the following field of views for the payload elements. The orbiter shall accommodate the sun exclusion fields of view and time of travel as negotiated in the ICDs.

The payload elements shall be accommodated on the orbiter such that the following fields of view are aligned with the specified orientation relative to Mars orbit reference coordinate system defined in 1.4-9.

Payload Element	**Field of View, degrees***	**Orientation**
PMIRR MK II	160 PMIRRinstr.step PMIRR160FOV.step	Nadir
MARCI+ Wide Angle detector Wide Angle stray light Medium Angle	 140 x 40 156 x 40 6 MARCIWACINST.step MARCIMACINST.step	 Nadir (140 deg aligned crosstrack) Nadir (156 deg aligned crosstrack) Nadir

	MARCIWACFOV.step MARCIMACFOV.step	
High Resolution Imager Detector Stray Light	 0.006 x 1.056 6.0 HRIINST.step HRI6FOV.step	 Nadir (1.056 deg aligned crosstrack) Nadir
Visible-Near Infrared Spectrometer	1.2 x 70 VNIRINST.step VNIRFOV.step	Nadir (70 deg aligned downtrack)
Shallow subsurface sounding radar	Omni directional	Antenna shall be in a plane perpendicular to Nadir
Electra Nadir UHF Antenna Zenith UHF Antenna X band LGA	 +/-60 +/-60 +/-60	 Nadir, Note 1 Zenith, Note 1 Nadir, Note 2
Op Nav Camera Detector Stray Light	 0.7, half angle, square +/- 10	 Note 3 Note 3

* = File names describe STEP files that contain CAD drawings of FOVs.

Note 1: The orbiter shall provide an environment such the multipath K-factor is at least 15 dB as measured at the surface of the UHF antenna for the specified FOV and that the multipath delay spread shall not exceed 10 nanoseconds.

Note 2: The orbiter shall provide an environment such the multipath K-factor is at least 15 dB as measured at the surface of the UHF antenna for the specified FOV.

Note 3: The Optical Navigation camera shall be accommodated such that the Martian moons Phobos and Deimos pass through its field of view during the last two weeks prior to MOI.

6.5. Serial Data Interface

The orbiter shall provide the low and high-speed serial interfaces to the payload elements as described in the following table. All serial interfaces from the orbiter to payload users shall be electrically isolated.

The orbiter shall support simultaneous real time data transfers from all payload elements to mass memory.

Payload Element	**Serial Interface**
PMIRR MK II	1 low speed
MARCI+	2 high speed
High Resolution Imager	1 high speed

Visible-Near Infrared Spectrometer	1 high speed
Shallow subsurface sounding radar	1 high speed
Electra	1 high speed
Op Nav Camera	1 high speed

6.5.1. High Speed LVDS

The orbiter shall provide high speed synchronous clock/data/enable serial data interfaces utilizing Low Voltage Differential Signaling (LVDS) compatible devices.

Each LVDS serial data interface shall be capable of supporting a data clock rate of up to 30 Mbps

Each LVDS serial data interface shall have at least 5 data clock rates of 30, 20, 10, 5, and 1 Mbps. The LVDS clock rate shall be commandable by orbiter software.

6.5.2. Low Speed RS-422

The orbiter shall provide low speed UART serial interfaces utilizing RS-422 compatible devices.

Each RS-422 data interface shall support a single data clock rate of 9600 bps.

6.6. Serial Command Interface

The orbiter shall provide synchronous clock/data/enable serial command interfaces utilizing RS-422 compatible devices. Each RS-422 command interface shall support a single data clock rate of 500 kbps.

6.7. Analog Data

The orbiter shall provide 4 analog temperature sensor channels for each payload element.

6.8. Digital Interface

The orbiter shall provide 2 programmable general purpose single ended digital I/O interfaces for each payload element. The digital I/O logic levels shall be 5V TTL compatible. All digital interfaces from the orbiter to payload users shall be electrically isolated.

6.9. Data Processing Capability

The orbiter shall provide a minimum of 20 MIPS processing capability for payload data processing, data compression, and command sequencing. The performance measure shall include the effects of processor system memory access and processor L1 cache misses, i.e. the processor system shall be matched so that effective performance of the CPU/Memory/Bus system will be 20 MIPS.

6.10. Time Services

The orbiter shall time tag payload data with an accuracy of +/-10 msec relative to the orbiter clock.

The orbiter shall distribute orbiter time once per second to any science instruments that require it, with an accuracy of +/- 1 ms.

A data word containing orbiter time shall be distributed, as well as a 1 Hz time tick. At each time tick, the most recently distributed value of orbiter time shall be valid.

6.11. Memory

The orbiter shall provide 48 Gbits of volatile mass memory for use by the payload.

To the extent practical the orbiter shall avoid erasing stored science data as a result of responding to on-board faults.

The orbiter shall be capable dynamically partitioning the payload memory based on the mission phase and the payload elements that are operating.

The minimum required effective data transfer rate from serial interfaces to mass memory shall be 100 Mbps.

The orbiter shall provide 10 Mbytes of non-volatile memory for use by the payload.

6.12. Command Storage

The orbiter shall provide at least 2 MB for payload element commands, plus timing information for each command.

6.13. Mounting

The instruments shall be mounted to the orbiter consistent with the accuracy requirements in section 6.16.

The MARCI+ medium angle, high-resolution imager and the visible-near infrared spectrometer shall be mounted such that their FOVs as defined in section 6.4 are aligned relative to each other within 0.5 mrad (3 sigma).

The orbiter shall enable the signal from the receive port of the X-band high gain antenna to be delivered to Electra with a maximum circuit loss of 2.5 db.

6.14. Thermal Interface

Unless specified otherwise in the instrument ICD, the payload elements shall be conductivity and radiatively isolated from the orbiter. The orbiter mounting surfaces for the payload elements, shall have an effective surface emissivity less than 0.1.

The orbiter shall maintain the temperature reference point of each instrument within the operating and non-operating ranges as specified in the instrument ICDs.

During normal primary science phase operations, the orbiter shall provide the Ultra Stable Oscillator [USO] in the Electra package with a temperature environment that changes no faster than 3° C/Hr in either direction.

6.15. Contamination Control

Contamination due to either the launch vehicle, the orbiter, or the environment shall not degrade the payload performance at any time during the mission beyond the levels negotiated into the payload ICDs.

Instrument purge lines and duration periods shall be as specified in the instrument ICDs.

6.16. Pointing Accuracy and Stability

The orbiter shall provide 3 axis pointing accuracy and stability as specified in the following tables. Accuracy includes all error contributions between the true stellar reference and each payload element's alignment reference. These accuracy requirements do not include the ephemeris errors in section 4.14, or the target location errors.

Pointing Accuracy

Axis	**Requirement (3 sigma)**
Roll	<0.7 mrad
Pitch	<1.0 mrad
Yaw	<1.0 mrad

Stability requirements shall be met during science observations as specified below. Stabilty is relative to the instrument mounting reference. It is not required to meet the stability requirements during reaction wheel momentum desaturations [if any], attitude slews, and during entrance and exit into/from solar eclipse.

Pointing Stability

Stability (3 sigma, per axis)		
Observation	**Duration**	**Requirement**
High Resolution Imaging	2 minutes	< 0.0015 mrad over 3 msec
VisNIR Imaging	5 minutes	< 0.05 mrad over 100 msec
MARCI+ Medium Angle Context Imaging	15 minutes	< 0.25 mrad over 1 sec
PMIRR Mk-II Atmospheric Sounding	Continuous	< 1.0 mrad over 2 sec and

		< 3.0 mrad over 16 sec

6.16.1. Cross Track Pointing

The orbiter shall be capable of pointing +/- 30 degrees in the cross track direction relative to the Mars orbit reference coordinate system.

After slewing to the cross track orientation, the orbiter shall meet the pointing requirements in 6.16.

The orbiter shall be capable of slewing to two different cross track orientations during the sunlit portion of one orbit at Mars.

6.17. Operations

The orbiter shall support payload operations as described below. Actual dates and the scope of the checkout and calibration activities will be determined for each instrument in its ICD.

6.17.1. Launch Phase

The payload will be launched in a powered off state.

6.17.2. Cruise Phase

The orbiter shall support the necessary operations to checkout and calibrate the payload. No payload operation shall be supported for the first fourteen (14) days after launch, during trajectory correction maneuvers, within thirty (30) days of Mars orbit insertion, or during other critical orbiter events.

6.17.3. Approach and Orbit Insertion

The orbiter shall support limited payload calibration and checkout activities in the MOI-60 days to MOI-30 days timeframe.

The orbiter shall support operation of the Optical Navigation Camera Experiment beginning at MOI-14 days and ending at MOI-2 days.

6.17.4. Aerobraking Phase

The orbiter shall support operations of the MARCI WA camera in support of aerobraking.

6.17.5. Primary Science Phase

The orbiter shall support simultaneous operation of all payload elements except the optical navigation camera experiment. The orbiter shall support the shallow subsurface sounding radar transmitter only during Sun eclipse and when the high power X band transmitter is off.

While nominally nadir pointing, the imaging instruments will require cross-track pointing as specified in 6.15.1.

The orbiter shall be capable of acquiring radiometric data via Electra from an in bound (to Mars) spacecraft using the high gain antenna during at least two 30-minute passes per day for 15 days. The angle between the direction to Earth and the direction to the in bound spacecraft will be less than 110 degrees. Normal science operations may be interrupted during these periods.

The orbiter shall not transmit an X band signal when Electra is in X band receive mode.

6.17.6. Relay Phase

The orbiter shall support the Electra package continuously and support limited operation of the science instruments.

The orbiter shall be capable of acquiring radiometric data via Electra from an in bound (to Mars) spacecraft using the high gain antenna during at least two 30-minute passes per day for 15 days. The angle between the direction to Earth and the direction to the in bound spacecraft will be less than 110 degrees.

The orbiter shall not transmit an X band signal when Electra is in X band receive mode.